IAA-L-1110

MULTIMISSION SPACE AND SOLAR PHYSICS MICROSPACECRAFT

David H. Collins

Manager, TAPD/NTPO Advanced Systems Technology Office

MS 301-490, Jet Propulsion Laboratory, California Institute of Technology

4800 Oak Grove Drive, Pasadena, California 91109-8099

ABSTRACT

The solar fields and particles environment and its interaction with planetary magnetospheres are not only of considerable scientific interest, they can impact human endeavors as well. Communications, power grids, and spacecraft operations can be disrupted by geomagnetic storms that are a consequence of sun-Earth interactions. Certain important processes are not well understood, and reliable warning of potential problems at Earth is usually less than an hour. Now, a detailed conceptual design exists for a mission that can both answer fundamental questions as well as provide hours-to-days of warning. Basically, a single launch sends nine microspacecraft to Venus where individually tailored gravity assists deploy them in specific solar orbits throughout a band covering 0.53 AU to 0.85 AU from the sun. From there they make the needed in situ observations. This paper discusses the mission context, what can be accomplished, cost minimization, the trajectories, early warning coverage, and the flight system.

INTRODUCTION AND BACKGROUND

The Problem

Particularly during "solar max" (a period of increased solar activity that occurs for roughly seven years out of eleven), huge, high-velocity plasma structures, called coronal mass ejections (CMEs), shoot outward from the sun, interacting with those things in their paths. Planetary interactions can be direct, though particle bombardment, and indirect, through storms caused in planetary magnetospheres. Many CMEs miss the Earth completely (but may not miss interplanetary spacecraft), and many others intercept the Earth but cause few, if any, problems. In fact, the nature and seriousness of these "space weather" effects varies radically depending on many factors, some of which are very poorly understood. The CMEs that do come by the Earth may take up to a few days to get here from the sun, and even if the CMEs are very large and energetic, the hazard they pose is quite dependent on their magnetic field orientations. In the more serious cases, though, CMEs result in hazardous high-energy particles in our environment and geomagnetic storms that cause disruptions in communications, surges in Earth power grids (and even blackouts), and interference and damage to spacecraft. For example, problems in space can include:

Surface and deep dielectric charging / Single event effects
Increased noise backgrounds for sensors / Solar cell damage
Ionization and displacement damage to other hardware
Hazard to humans in space (and in aircraft on polar flights)

Current Hazardous Space Weather Forecasting Limitations

No good method has been found for predicting when a CME that is hazardous to our interests will form and erupt from the sun. There are, however, at least two methods that can give us particularly important information about a CME after it leaves the sun. One method is to remotely observe the region between the sun and Earth from two or more spacecraft that are positioned to have good perspectives for both seeing and tracking a CME in the region. This can provide a particularly early indication that a CME is headed toward Earth, an estimate of expected arrival time, and some idea of CME strength. Unfortunately, spacecraft positions that both satisfy these conditions and that are relatively easy to maintain are outside the region and do not accommodate measurements within the CME. This leaves in doubt key information about particle fluxes and field strengths and directions. In a second method, the focus is on making the in situ measurements. In this case, one or more spacecraft between the sun and Earth intercept any CME traveling toward Earth, make key measurements, and, when needed, warn the Earth. In addition, they can provide particularly important information about the nature and seriousness of the threat. The primary limitation in assessing the threat here is not inherent in the method but in its current implementation. This implementation makes clever use of the Earth-sun L1 libration point to station a spacecraft continuously off the sun side of Earth and very near to the sun-Earth line. Unfortunately, the sun is only 1% closer to this point than it is to Earth. This limits warning time to approximately an hour or less and leaves little time to confirm a threat or study its characteristics. A further problem with both this implementation of the in situ method and the remote sensing method is that neither can answer certain important questions about CME processes, structure, and evolution. What would be desirable is to have the advantages of the in situ method without the L1 distance limitation.

Directly Related Previous Studies

A revolutionary vision and approach for a new generation of spacecraft was developed by NASA/JPL in 1993-1995 along with example flight system concepts that were consistent with that approach and helped illustrate its potential[1]. One of these[2] was the initial concept behind the study discussed here. In this concept, the spacecraft was capable of several different missions, but the most important, and primary design driver, was a single-spacecraft solar early warning precursor mission with a 0.5 AU perihelion (where the sun is 50% closer than the sun-Earth distance). Subsequently, a set of various microspacecraft studies[3,4] was jointly carried out by JPL and the Air Force Research Laboratory between the summer of 1998 and April 1999. One very small study started with the earlier concept[2], developed trajectories for a multiple-spacecraft early warning mission, reduced dependence on certain very long-range new technology developments, and incorporated some updates based on recent work.

The MSSPM Approach and Benefits

The much larger NASA/JPL study discussed in this paper started with the April 1999 results but focused on much deeper analysis of what can be accomplished, the flight system (including its payload), trajectories and spread of spacecraft around the sun, and launch, injection, and deployment. In the basic concept it places nine microspacecraft in unique complementary orbits around the sun that collectively provide the needed coverage. The orbits are carefully designed to insure that nearly continuously at least one of the microspacecraft is close to the sun-Earth line and much closer to the sun than the sun-Earth distance. Specifically, the largest aphelion distance is 0.85 AU and the smallest perihelion distance is 0.53 AU. So, the sun is roughly 15 to 47 times closer to spacecraft in this region than it is to L1, and hours to days of early warning is available. Another MSSPM benefit is that it can answer important questions about the physics of space weather. This benefit is derived from the wide distribution of microspacecraft around the sun that allows interception of most large CMEs (not just those headed toward Earth) by multiple microspacecraft at various solar distances and over a broad range of solar clock angles.

OBJECTIVES

As in the previous studies, MSSPM is capable of multiple missions, but the one of highest interest and the focus in this study is the solar early warning mission. Its primary objectives include:

Better understanding of the physics of hazardous space weather, particularly particle acceleration at shocks and the large-scale structure and radial evolution of CMEs

Forecasting hazardous space weather with hours-to-days warning of severe geomagnetic activity, high energetic particle peak fluxes, and the possibility of killer electrons in the magnetosphere

APPROACH

Since this mission utilizes a relatively large number of spacecraft, it is particularly important to use an approach that holds down spacecraft, launch, and operations costs. To do this, the approach utilized here incorporates many of the elements of the approach developed in 1993-1995[1], and this implies an aggressive use of new technology. While that technology development is likely to require some funding augmentation in certain cases, the net result is expected to be both lower cost for MSSPM as well as lower costs for many subsequent missions. Elements of the approach used here can be briefly summarized as follows:

> Make the spacecraft identical or nearly identical with each other to move into a recurring cost regime and reduce spacecraft costs.
>
> Minimize spacecraft size and mass to reduce launch costs, in particular, as well as costs for facility use, assembly, and transportation.
>
> Make use of on-board data analysis/compression and autonomous control to both reduce telecommunications hardware size, mass, and cost and to reduce ground operations and data analysis costs.
>
> Minimize needed flight system resources and complexity, specifically:
>
> > Use a small, focused payload to reduce spacecraft size, mass, and cost and to help reduce ground operations and data analysis costs.
> >
> > Minimize the size, mass, and power needs of spacecraft components and assemblies to help reduce spacecraft size, mass, and costs.
> >
> > Avoid the use of nuclear fuel and other hazardous materials to improve personnel safety and help reduce launch approval costs.
> >
> > Only incorporate redundancy where it is most cost effective and thus help reduce spacecraft size, mass, and cost.

MISSION / TRAJECTORIES / COVERAGE

The mission uses a single launch vehicle to send multiple spacecraft toward Venus where it uses customized Venus gravity assists (VGAs) at different altitudes to help pull the microspacecraft into different solar orbits well inside 1 AU. This causes the microspacecraft to slowly spread out around the sun (and to have constantly changing geometries with respect to the sun, Earth, and each other). The resulting band of microspacecraft around the sun allows multipoint scientific measurements within most large CMEs, regardless of radial direction, and provides CME early warning coverage for Earth. Optionally, a second launch near the time of the first could be used to send a second group of microspacecraft to Venus on an intentionally slower trajectory. At Venus, these would be dispersed into different, complementary solar orbits but this time from a different Venus position. This would both increase the sampling density and broaden the coverage of the collection of orbits. Launch opportunities come approximately at 1.6-year intervals, and it is thought that a set of equally good solar orbits can be found for each opportunity. Based on new technology development and solar cycle phasing considerations, the 2005 launch opportunity is probably the earliest that should be used, and it is the baseline for this study. The 2007 opportunity would allow more time for technology development and probably fits better with the solar cycle. Although the next opportunity after 2007 would allow still more technology development time, both the early warning and broad scientific coverage would become operational somewhat later in the solar cycle than is probably desired.

Specific details of the baseline are as follows. Nine microspacecraft are launched on a small, lower cost, Taurus-class vehicle with an upper stage added for the interplanetary injection. The nominal launch date is 10/29/05, and a Type 1 trajectory with C3=13.28 (km/sec)^2 and declination=11.70° is used with Venus flybys on 2/12/06. Immediately after injection on the trajectory to Venus, the spacecraft separate and fly close but independently to Venus where tailored VGAs are used to send each into the particular, unique solar orbit identified in the table below. (Calculated nominal values are shown, but the full precision displayed is not required operationally.) The optimized orbits are entirely within 0.85 AU of the sun, and at least one microspacecraft is within ±22.5° of the sun-Earth line (and available for early warning coverage) better than 92% of the time during the 1900-day period considered.

Period (months)	Radius of Perihelion (AU)	Radius of Aphelion (AU)	Inclination (degrees)	Node (degrees)	Argument of Periapsis (degrees)	VGA Altitude (km)
6.10	0.53	0.75	4.47	31.1	-85.4	8147
6.30	0.57	0.79	4.63	28.8	-89.72	10726
6.70	0.58	0.77	4.83	26.4	-102.16	16866
6.82	0.59	0.78	4.86	26.1	-106.5	19051
7.12	0.61	0.80	4.89	25.7	-118.5	25495
7.30	0.63	0.81	4.88	25.8	-125.9	30248
7.43	0.63	0.82	4.87	26.0	-131.3	34237
7.63	0.64	0.84	4.83	26.4	-139.3	41574
7.80	0.65	0.85	4.79	26.9	-145.8	49356

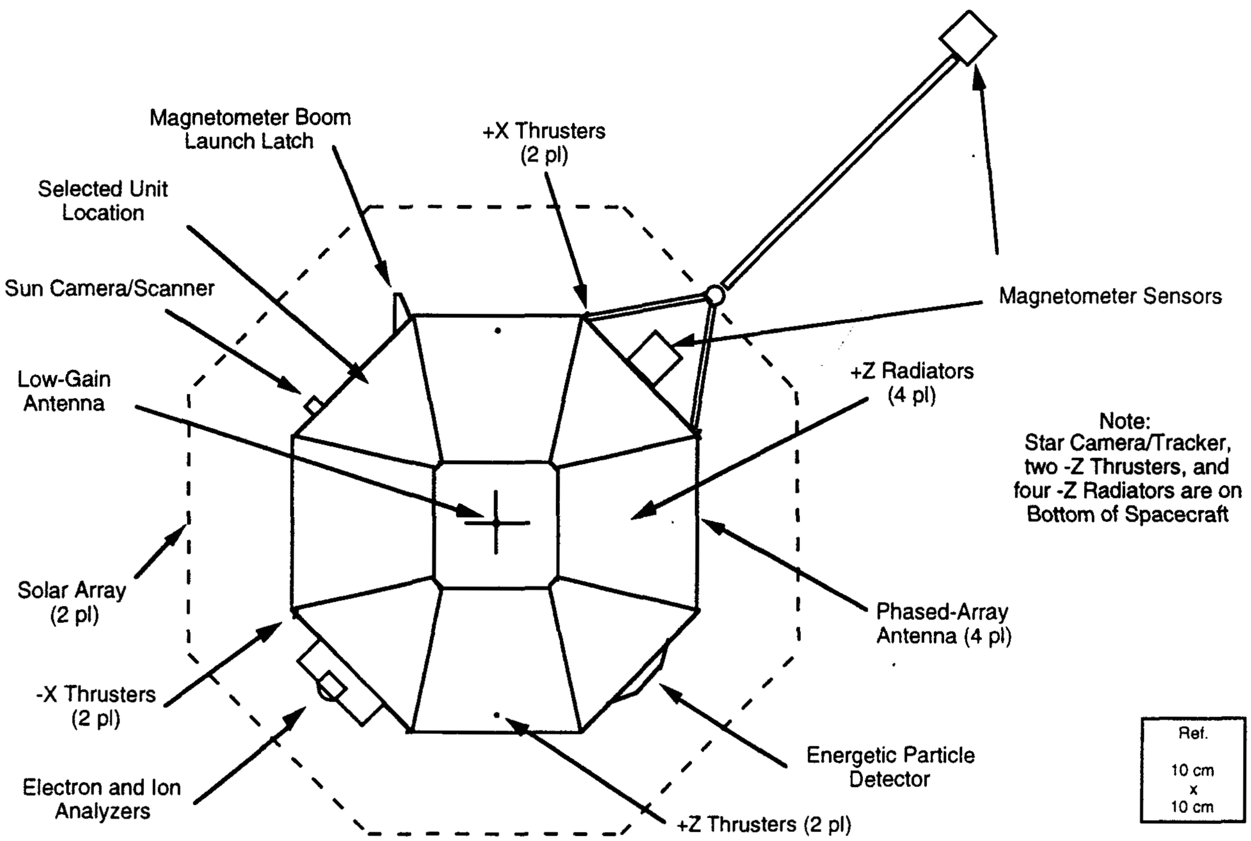
FLIGHT SYSTEM

Key Interrelated Technical Drivers and Mass Summary

Fitting nine microspacecraft, their integration and separation system, and an upper stage into the volume and performance capability of a small, lower-cost, launch vehicle is a key driver on the flight system design and particularly on the size and mass of the microspacecraft. Once launched, injected, and separated, an early trajectory correction maneuver (TCM) to null out possible launch and injection errors places the fundamental demands on flight system ΔV capability and does so when available power is at a minimum. (Associated with both this maneuver and the later, and much lower ΔV, Venus targeting maneuver is the need for the flight system to support the necessary navigation.) Optional scientific measurements can begin even on the way to Venus, before the microspacecraft are in their operational solar orbits, and the required measurements begin after VGA. The need to make these physical measurements, as well as those for guidance and control, within the size, mass, and power resources of the spacecraft is another key driver on the flight system. Also within these quite limited resources, the flight system must process the data and provide communications with Earth. An additional, and particularly important, diver is the need for the flight system to be able to operate over the one-to-four suns equivalent thermal environment (as well as the radiation environment within the inner solar system). A detailed conceptual design for a flight system that meets all these needs has been developed and is briefly summarized below. The mass for each spacecraft is 15 kg, which includes 35% contingency and both ΔV and attitude control propellant.

Configuration

The microspacecraft looks somewhat like a bobbin as did one of its predecessors to the inner solar system (for similar reasons), the Helios spacecraft. The drawing below shows a top view of the spacecraft with an X-ray view through its top solar array. The microspacecraft core is octagonal with approximate dimensions of 40 cm from side to side by 12 cm thick. The spin axis extends through and is perpendicular to the octagonal surfaces. Extending upward and outward from the perimeter of the top core surface with a 45° cone angle is a semi-conical solar array with eight trapezoidal solar panel areas. A second solar array, a mirror image of the first, extends downward and outward from the bottom core surface perimeter. (Solar cells on these arrays are on the radially outboard sides.) The outermost edges of each solar array are 60 cm apart, and the basic height of the microspacecraft is 32 cm. This excludes a 0.6-cm diameter low-gain antenna and its support tube that extend upward from the center of the top core surface to a point 21 cm above the top of the top solar array. A cylindrical, 3.6-cm diameter star camera/tracker followed by its 6-cm diameter baffle extend downward from the center of the bottom core surface 8 cm and are shielded by the solar arrays from the sun. Four of the rectangular faces of the core are phased-array antennas, and these faces alternate with faces that support fields and particles sensors and a sun camera/scanner. Since the microspacecraft rotates with its spin axis perpendicular to the sun-spacecraft-Earth plane, the radials through these faces sweep through both the sun and Earth.



Payload

The focused payload of instruments shown in the drawing covers the measurements of primary importance for this mission and has energy ranges that support both the scientific and warning objectives. The energetic particle detector uses the dE/dx vs. total energy technique in a particle telescope with silicon detectors. The electron and ion analyzer uses a dual "top hat" design. And the magnetometer uses two triaxial fluxgate sensors. In addition, a constrained accommodation is provided for a selectable microinstrument in the bay with the sun camera/scanner, and it can look out that face of the microspacecraft. One candidate for this microinstrument is an EUV monitor. Those microspacecraft that do not include the optional microinstrument add a supplemental secondary battery instead, increasing energy storage capability by ≈40%.

Power

Triple-junction GaAs solar cells provide a minimum of 31 W near Earth and much more later, and a lithium-ion polymer battery provides a base energy storage capability of ≈44 Wh.

Information Processing and Control

Even though the basic digital processing needs of the flight system are low, a substantial processing capability is provided to enable on-board data analysis/compression plus increased autonomy. A 16-MByte flash ROM provides non-volatile storage for the operating system, critical engineering and scientific algorithms, and critical parameters and constants. A 64-MByte RAM with error detection and correction provides working memory as well as data storage between downlinks to Earth. And a microprocessor provides greater than 200-MIPS processing.

Telecommunications

A 2-dBi, X-band antenna with a toroidal pattern about the spin axis is used for uplink reception, and four 19-dBi, X-band phased-array antennas are used for the downlink. Output from a 6-W_{RF} amplifier is sequentially sent to the four antennas as their boresights come within 45° of Earth, and they then hold the beam on Earth. A transponder provides receiver, exciter, modulator, navigation, and beacon tone generation functions. Downlink performance with periodic tracks from a 34-m beam waveguide deep space network (DSN) antenna for science return varies over ranges of 4 - 7 kb/s at 0.2 AU and 4 – 64 b/s at 1.8 AU, depending on what DSN upgrades are incorporated by the time of the mission. Continuous beacon-warning monitoring of whichever microspacecraft happens to be closest to the sun-Earth line is provided by the DSN 11-m network with additional hardware that supports both beacon alert and beacon data return modes.

Guidance and Control

The microspacecraft is spin stabilized and uses the radial sun camera/scanner and axial star camera/tracker as its celestial attitude references. Active nutation damping with thrusters is used during ΔV maneuvers, and passive nutation damping is used at other times. A 3-axis microgyro provides an inertial reference and helps in both the initial attitude acquisition phase after launch/injection and in controlling the active nutation damping. A 1-axis microaccelerometer helps in determining ΔV magnitude during maneuvers.

Propulsion

Liquid ammonia propellant is stored in four composite-overwrapped tanks and is metered into a vaporizer as needed. A plenum buffers vapor use by eight 5-mN thrusters, which are arranged in couples and provide 3-axis control during acquisition, spin rate and spin axis orientation control, and ΔV in lateral and axial directions.

Packaging and Structure

An electronics box with 0.25-cm thick aluminum walls is at the center of the microspacecraft. It houses the critical electronics and helps to isothermalize them and shield them from radiation. It also supports thin battery cells on four sides and the inboard ends of the propellant tanks. Eight radiators are bolted to flanges at the top and bottom of the box with each radial pair partially enclosing a propellant tank. Semi-radial structure at the box edges in combination with the radiators and faces of the microspacecraft core complete its primary structure.

Temperature Control

The microspacecraft configuration shades the radiators and star camera/tracker, and spinning helps lower side temperatures and isothermalize the interior. (Solar array temperatures are further lowered due to the outward tilt of the arrays.) Solar heat shields on the sensor bays, thermal white paint patterns, and the use of highly polished aluminum control the heat absorptance and emittance from surfaces. And fiberglass semi-radial and side structure and 10-layer thermal blankets greatly reduce unwanted heat transfer. Electrically controlled radiator surfaces increase emittance of the radiators to offset extra heat dissipation when the transmitter is used, and heaters further limit internal temperature excursions when the transmitter is off.

CONCLUSION

The mission and flight system are technically feasible, would greatly expand knowledge of the physics of space weather, and would provide hours-to-days warning of hazardous space weather.

REFERENCES

1. Collins, D., Kukkonen, C., and Venneri, S., "Miniature, Low-Cost, Highly Autonomous Spacecraft -- A Focus for the New Millennium," 46th International Astronautical Congress, IAF-95-U.2.06, Oslo, Norway, Oct. 1995

2. Collins, D. H., Horvath, J. C., et al., "Space Physics Fields and Particles Mission Class Spacecraft," *Second Generation Microspacecraft Final Reports for 1994*, JPL internal document D-12645, Rev. A, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, CA 1995, pp. 59-89

3. Collins, D., Pomphrey, R., Yen, C., Stallard, M., Cooke, D., et al., "Multimission Space and Solar Physics Microspacecraft," *Air Force Research Laboratory -- Jet Propulsion Laboratory Future Collaborations in Microsatellites Study*, AFRL-JPL internal document, Pasadena, CA, Apr. 1999

4. Moser, R., Collins, D., et al., "Novel Missions for Next Generation Microsatellites: The Results of a Joint AFRL-JPL Study," 13th Annual AIAA/USU Conference on Small Satellites, SSC99-VII-1, Logan, UT, Aug. 1999

ACKNOWLEDGEMENTS

The research in this document was carried out by the Jet Propulsion Laboratory (JPL), California Institute of Technology under a contract with the National Aeronautics and Space Administration (NASA). This research was performed for the Advanced Systems Technology Office of the JPL Technology and Applications Programs Directorate (TAPD) by personnel from the JPL Engineering and Science Directorate. The research was sponsored by the NASA Space Science Office and supported within JPL by the NASA Technology Program Office (NTPO).

Contributors* to this research included: Jim Alexander, David Bame, Doug Bernard, Gaj Birur, Willard Bollman, David Collins, David Cooke (AFRL), Don Croley, Mike Davis, Pat Dillon, Keith English, Wai-Chi Fang, Joan Feynman, Gus Forsberg, Warren Frick (OSC), Jevan Furmanski, Bob Glaser, Bruce Goldstein, Ray Goldstein (SWRI), Richard Grumm, Peter Guggenheim (VTI), Martin Herman, John Huang, Anil Kantak, Ken Kelly, Mike Lara (TC), Alan Lazarus (MIT), Joe Lewis, Paulett Liewer, Steve Macenka, Warren Martin, Dave McGee, Dick Mewaldt (CIT), Anthony Mittskus, Bill Moore, David Morabito, Guenter Musmann (TUB), Barry Nakazono, Tim O'Donnell, Henry OuYang (AC), Jonathan Perret, Rick Pomphrey, Wayne Pryor (LASP), Tom Shaw, Dennis Shebel, Mike Stallard (AFRL-AC), Paul Stella, Carlo Stjuste, Steve Swift (CWSA), Kent Tobiska, Bruce Tsurutani, Glenn Tsuyuki, Al Whittlesey, Mark Wiedenbeck, Daniel Winterhalter, Tom Woods (LASP), Chen-wan Yen, Sam Zingales.

---

* The affiliations of non-JPL contributors are indicated as follows: AC, Aerospace Corporation; AFRL, Air Force Research Laboratory; CIT, California Institute of Technology; CWSA, C W Swift & Associates, Inc.; LASP, Laboratory for Atmospheric and Space Physics; MIT, Massachusetts Institute of Technology; OSC, Orbital Sciences Corporation; SWRI, Southwest Research Institute; TC, Thiokol Corporation; TUB, Technical University of Braunschweig, VTI, Valence Technology, Inc.